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REYNOLDS NUMBER REQUIREMENTS FOR VALID TESTING AT TRANSONIC SPEEDS

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SUMMARY

The variation of wing shock location with Reynolds number has been examined for configurations for which both flight and wind tunnel wing pressure distribution data were available. The purpose of the study was to determine if there is a minimum level of Reynolds number, short of full scale, at which reliable flow simulation can be achieved in transonic test facilities.

The shock locations as a function of Reynolds number at conditions of constant Mach number and angle of attack were normalized so that shock position was obtained in relative terms from zero to one for each configuration and condition studied. Normalizing the shock location permitted the comparison of data for different configurations and conditions on a common basis. An implicit thought in the approach used was that while no one set of data may be considered definitive, if enough data were studied then some consistent pattern might emerge.

Not enough data has been analyzed thus far to obtain conclusive results. However, preliminary indications are that a substantial increase in Reynolds number capability may be required in future transonic test facilities if test procedures cannot be developed for adequate flow simulation within existing facility capabilities.

NOTATION

C_p	pressure coefficient
c	wing chord, meters
K	constant
M	Mach number
P	exponent
R	Reynolds number based on wing chord
x	streamwise distance measured from wing leading edge, meters
x_A	most forward location which shock apparently approaches asymptotically at low Reynolds numbers, meters
x_B	most rearward location which shock apparently approaches asymptotically at high Reynolds numbers, meters
x_S	shock location, meters
x_S'	normalized shock location, $x_S' = \dfrac{x_S - x_A}{x_B - x_A}$
α	angle of attack, degrees

1. INTRODUCTION

One of the requirements for valid aerodynamic testing at transonic speeds is that the test Reynolds number be sufficiently high so as to permit adequate duplication of shock-boundary layer interaction and separated flow effects. Intensive efforts are currently underway in several aerodynamic research centers to develop methods of correctly simulating full scale transonic flow effects within the Reynolds number capability of current research facilities. General methods are not yet known for correct flow simulation which can be applied on a routine basis compatible with the usual aerodynamic test requirements for most configurations, although some notable success has been achieved with special techniques in specific instances (see, for example, reference 1). If these efforts do not result in test procedures in which a high level of confidence can uniformly be placed, then it appears likely that testing at substantially higher Reynolds numbers will be necessary. The purpose of this investigation was to determine if there is a minimum level of Reynolds number, short of full scale, at which reliable flow simulation can be achieved in the absence of special techniques. This information is of interest in evaluating current research facility capabilities and in defining the size, pressurization, and power requirements for future test facilities.

A number of familiar aerodynamic problems associated with separated flows are known to be affected by testing at reduced or model scale Reynolds numbers with fully turbulent boundary layers at transonic speeds. The following affected problem areas may be considered typical: (a) forward location of shock on an airfoil chord, (b) creeping drag rise and/or decrease in drag rise Mach number, (c) decrease in lift coefficient for pitching moment linearity break (pitch-up), (d) decrease in lift coefficient for buffet onset. These adverse effects of testing at reduced Reynolds number occur or are accentuated primarily because the boundary layer does not remain geometrically scaled as Reynolds number is decreased from full scale to model scale. This lack of scaling was pointed out by Loving in reference 2 where a large discrepancy in shock location between wind tunnel and flight test results was attributed to a relatively thicker boundary layer at model scale Reynolds number.

In attempting to determine the minimum level of Reynolds number for valid transonic testing, the variation of wing shock location with Reynolds number has been selected for study. This selection was made because shock location was considered to be a major contributing factor to the aerodynamic behavior observed in the other three problem areas noted above. In addition, the shock location data were considered to be the best defined and most reliably determined measurements available for study from existing data sources.

2. METHOD OF ANALYSIS

2.1 Assumptions

The following assumptions are inherent either explicity or implicity in the approach used in the shock location study. (1) While no one set of data was considered definitive, it was assumed that if enough data were studied then some consistent pattern might begin to emerge. (2) The shock location was assumed to be a primary indicator of the sensitivity of the shock-boundary layer interaction to Reynolds number. (3) A terminal shock position was assumed to exist in the limit of increasingly large Reynolds number. (4) Mach number, angle of attack and Reynolds number were assumed to be the primary variables affecting shock position. The last assumption is in effect a consequence of ignoring differences between wind tunnel and flight test conditions. These differences can be caused by but are not limited to the effects of wall constraint and model support interference, airstream turbulence, model dynamics, surface roughness, and propulsion simulation or the lack of it.

2.2 Procedure

The shock locations were determined from pressure distributions on the upper surface of wings or airfoil sections. The shocks were assumed to be located in the positive pressure gradient immediately behind the crests in the pressure distributions as shown in the upper sketch of figure 1. Shock locations were determined in this way at fixed conditions of Mach number and angle of attack over as large a range of Reynolds number as possible using both flight and wind tunnel data sources. The wind tunnel data were for the condition of boundary layer transition artificially fixed near the leading edge. Natural transition is assumed to have occurred near the leading edge in the flight tests.

The variation of shock location with Reynolds number tended to appear as shown in the lower sketch of figure 1. The asymptotic behavior of the shock location with increasing Reynolds number (x_S approaches x_B for R on the order of 10^8) might be expected since the inertial forces increasingly predominate over viscous forces and the flow tends to approach the inviscid limit. The sketches in the addendum to reference 3 indicate a similar behavior of shock location as the Reynolds number grows very large. Theoretical inviscid flow solutions should be valid as this limiting condition is approached. The asymptotic behavior of the shock location with decreasing Reynolds number (x_S approaches x_A for R on the order of 10^6) is less predictable but is nevertheless well documented by the available data. It should be noted that this behavior has only been observed in the Reynolds number range from 10^6 to 10^8 for fully turbulent boundary layers.

For each set of data, that is, shock locations as a function of Reynolds number at a constant Mach number and angle of attack for a given configuration, the shock locations were normalized so that the shock location from the most forward to the most rearward positions was obtained in relative terms from zero to one for any given set of data. The normalized shock location was obtained from the following relation:

$$x_S' = \frac{x_S - x_A}{x_B - x_A}$$

where x_B and x_A are the upper and lower asymptotes respectively.

Normalizing the shock location is advantageous in that it allows sets of data from different sources to be compared on a common basis. It has the disadvantage that it requires extensive Reynolds number coverage for any given set of data in order to establish the upper and lower asymptotes.

3. RESULTS

In the study thus far, wing shock location data from two configurations have been analyzed; the C-141 shown in figure 2 and the F-80 shown in figure 3a. The wind tunnel data for the F-80 wing section were obtained from a two-dimensional model shown in figure 3b. The wind tunnel tests on this model were conducted in the NASA Langley 8-foot transonic pressure tunnel by A. A. Luoma. The data are as yet unpublished but are similar to the data presented by Blackwell in reference 1. The F-80 airplane flight data were obtained from reference 4. The C-141 airplane flight data were obtained from reference 5 and the corresponding wind tunnel data were obtained from reference 6.

The normalized shock locations were plotted in the form $\log \dfrac{x_S'}{1 - x_S'}$ versus $\log R$ as shown in figure 4. Most of the data is seen to fall within the shaded band. A linear variation of the data on figure 4 would imply a normalized shock location as a function of Reynolds number in the form

$$x_S' = \frac{R^P}{R^P + K^P}$$

With the data replotted as x_S' versus $\log R$ as shown in figure 5, the straight shaded band of figure 4 is seen to change into the S-shaped band shown on this figure.

It should be pointed out that all of the experimental shock location data used in the present analysis are subject to some amount of uncertainty. When the shock locations are normalized, these uncertainties tend to become magnified and thus account for much of the scatter evident in the data of figure 5. In addition, some care must be exercised in combining data from wind tunnel and flight sources in order to insure that comparable conditions are obtained. The angle of attack in flight and the differences in aeroelastic deflection between the wind tunnel and flight test vehicles are especially important factors in this regard and are sometimes difficult to determine accurately. An increase in the amount of data analyzed should increase the reliability of the trends which these data tend to exhibit.

On the basis of the limited amount of data analyzed thus far, a preliminary indication is that the Reynolds number range which would cause the shock to be located within 90 percent of its rearmost or terminal position is of the order of thirty to forty million. However no firm conclusions can be drawn at this time.

4. CONCLUDING REMARKS

In the foregoing analysis, the location of a shock on the upper surface of an airfoil is taken as an index of the severity of the effects of shock-induced separation as influenced by Reynolds number variation. No attempt has been made here to differentiate between flows that are designated as model A and model B flows in reference 3. Model B flows are described in reference 3 as those where trailing edge separation plays a significant role in the shock-boundary layer interaction and its attendant flow separation. Model A flows are those where trailing edge separation either is not present or does not significantly affect shock induced separation.

The results obtained thus far are not conclusive but tend to indicate that a substantial increase in Reynolds number capability may be required in future transonic test facilities if alternative methods of correct flow simulation cannot be successfully developed.

REFERENCES

(1)	Blackwell, James A.,Jr.	Effect of Reynolds Number and Boundary-Layer Transition Location on Shock-Induced Separation. Paper presented at a Specialists' Meeting of the Fluid Dynamics Panel of AGARD (Paris, France), Sept. 18-20, 1968.
(2)	Loving, Donald L.	Wind-Tunnel-Flight Correlation Of Shock-Induced Separated Flow. NASA TN D-3580, 1966.
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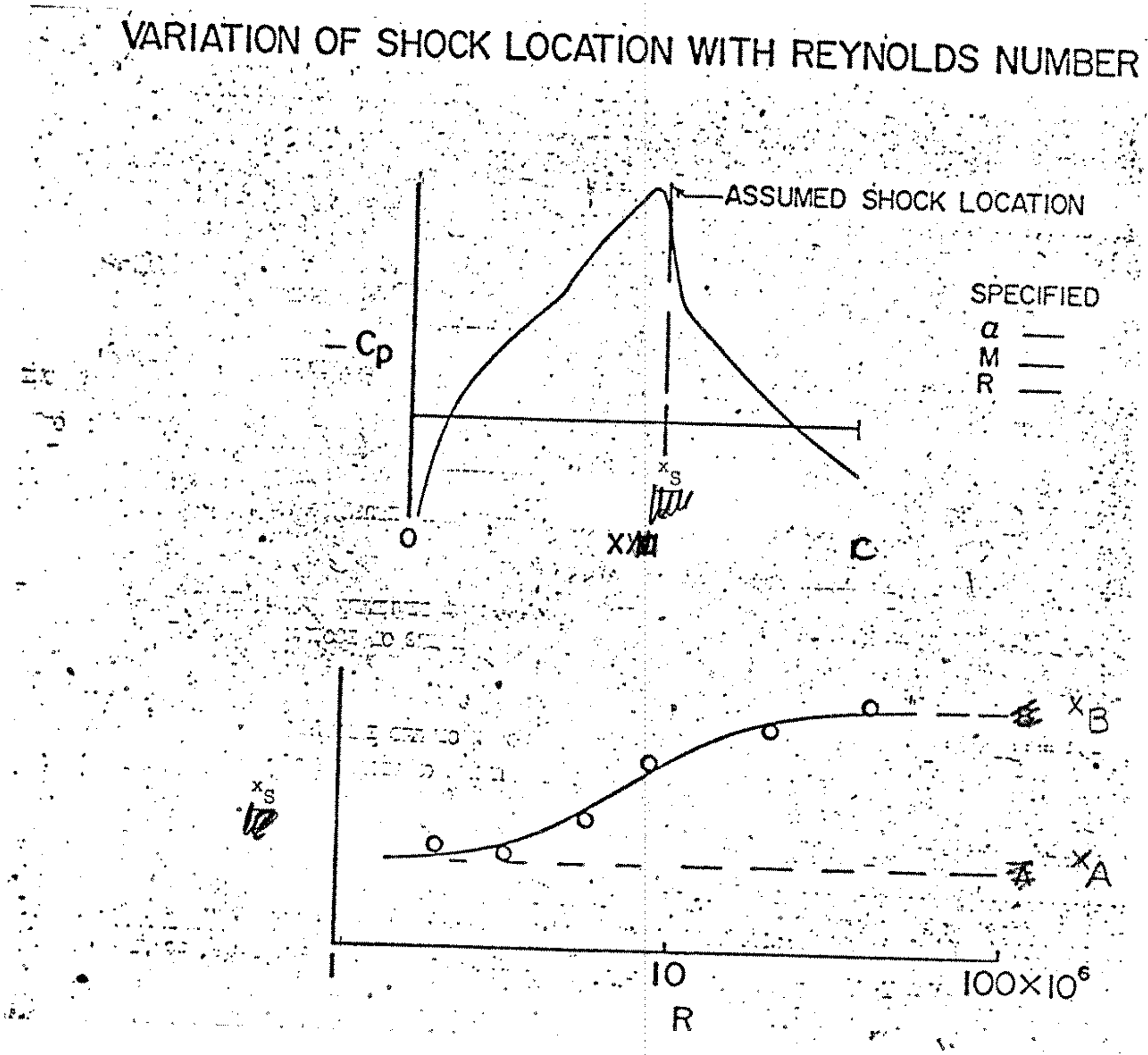


Figure ~~3~~ 1.- Variation of shock location with Reynolds number.

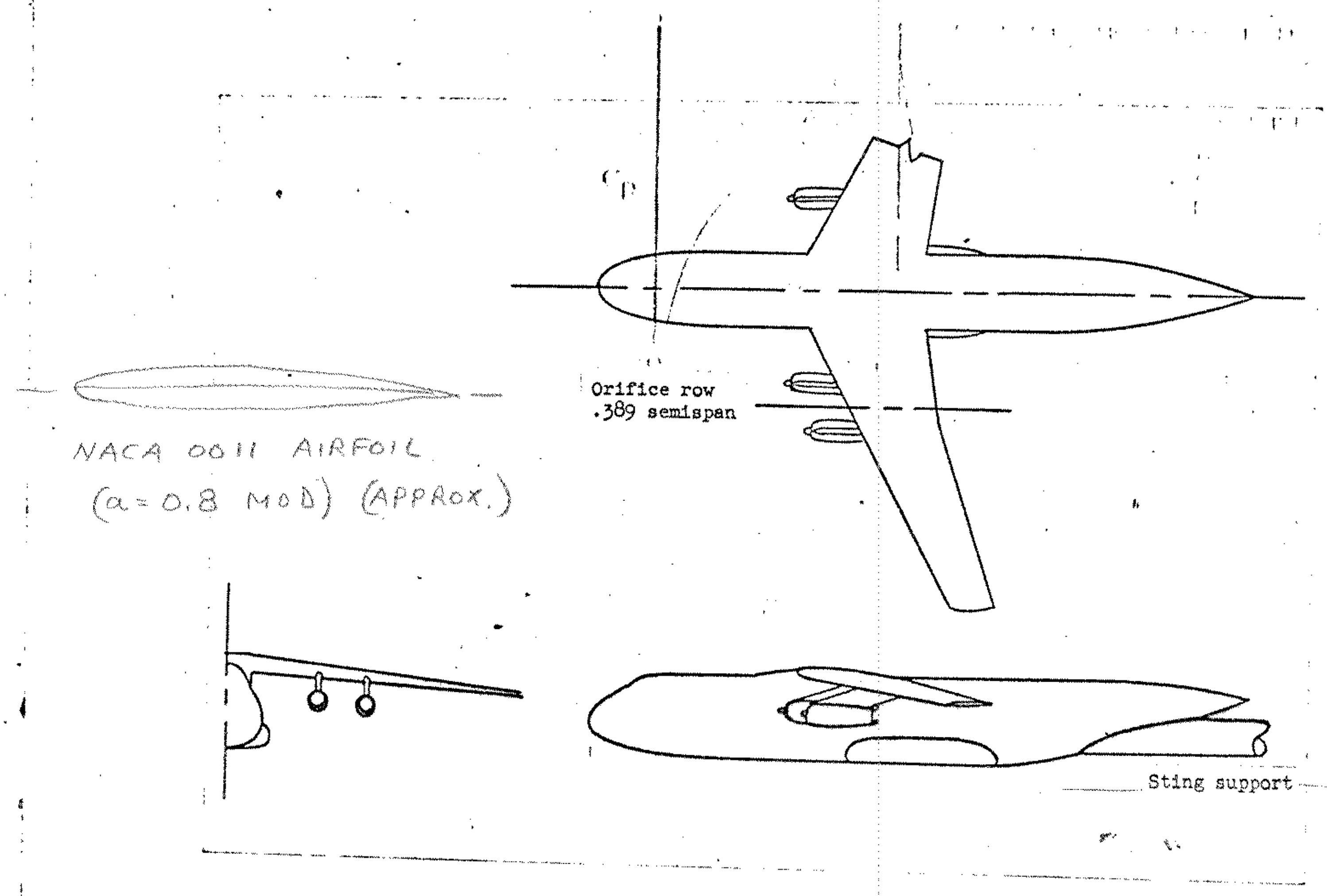
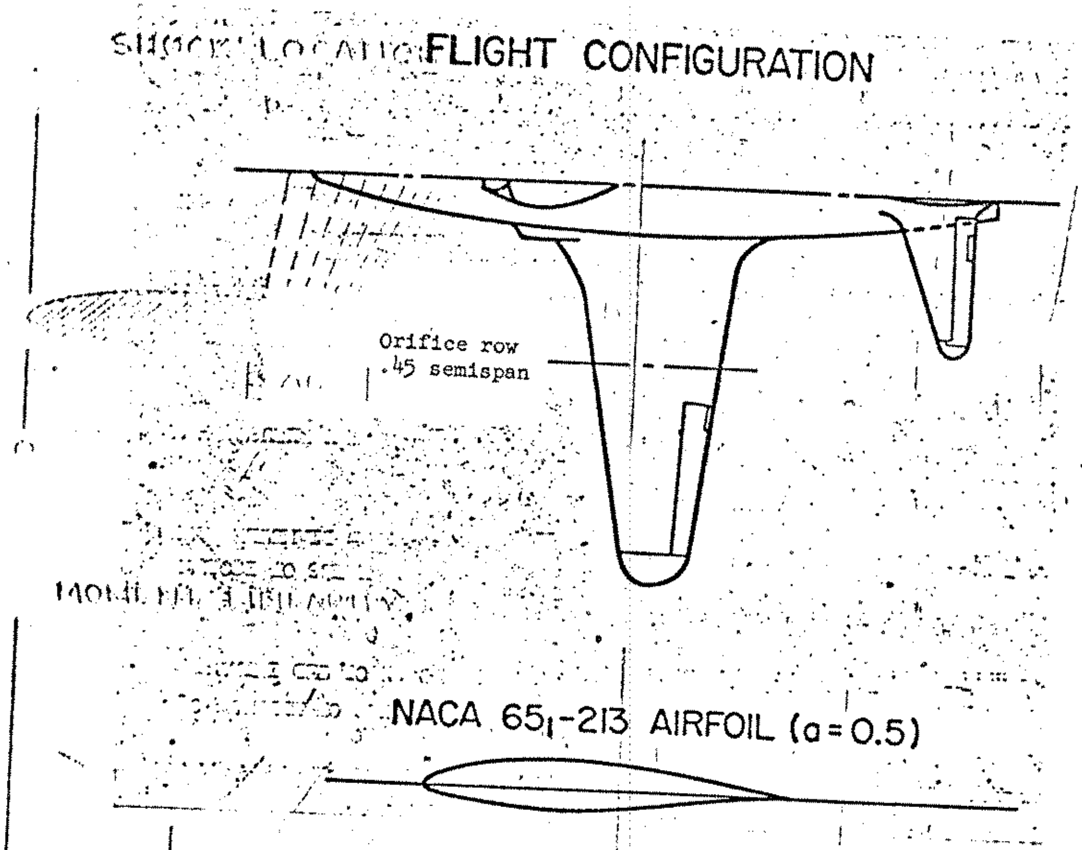
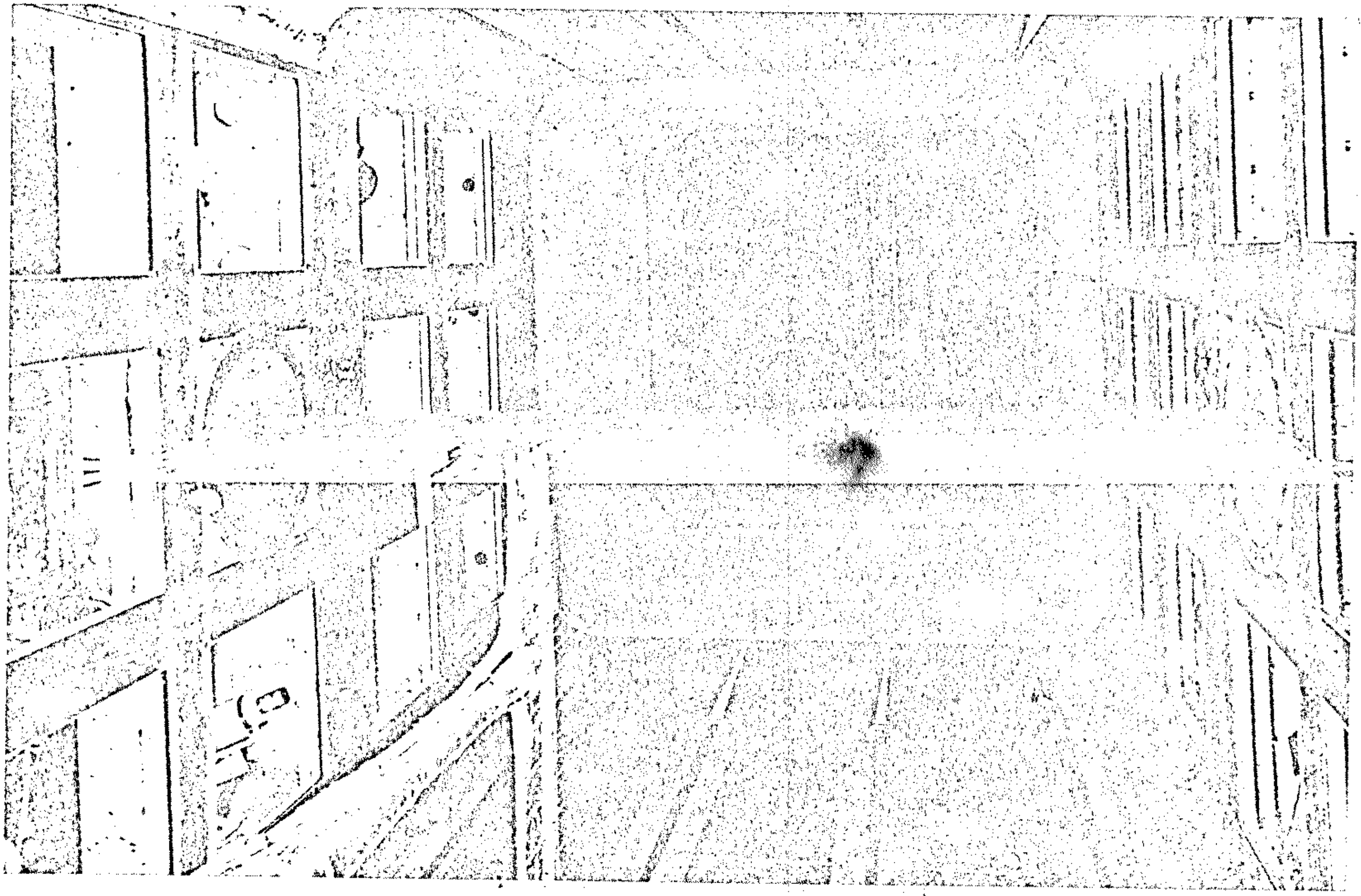


Figure ~~1~~ 2.- C-141 wind tunnel model configuration.



(a) F-80 airplane.

Figure 3.- F-80 flight and wind tunnel configuration.



(b) Two-dimensional wind tunnel model of F-80 airplane wing section.

Figure ~~5~~ 3.- Concluded.

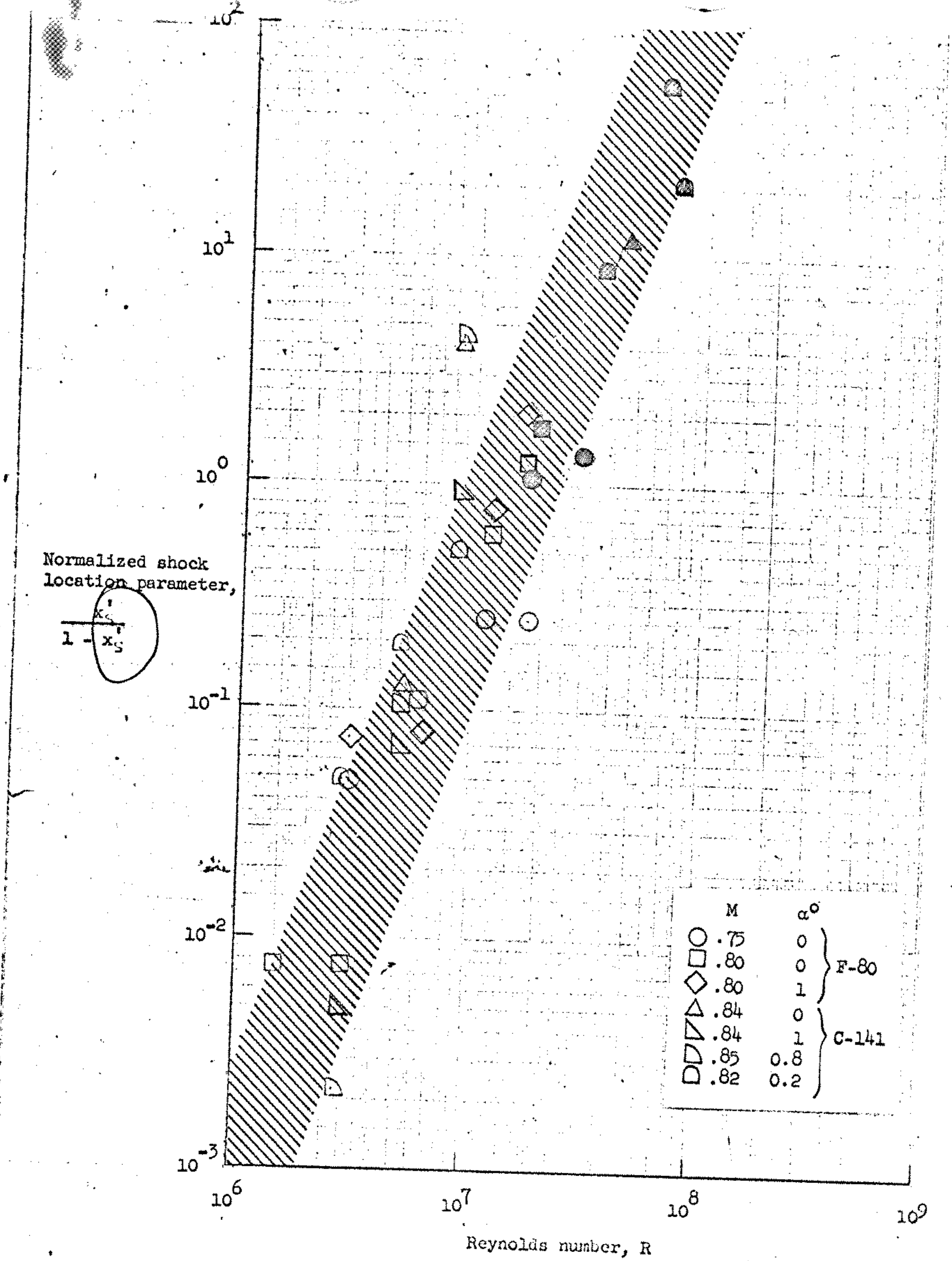


Figure 4.- Normalized shock location parameter as a function of Reynolds number (solid symbols indicate flight data).

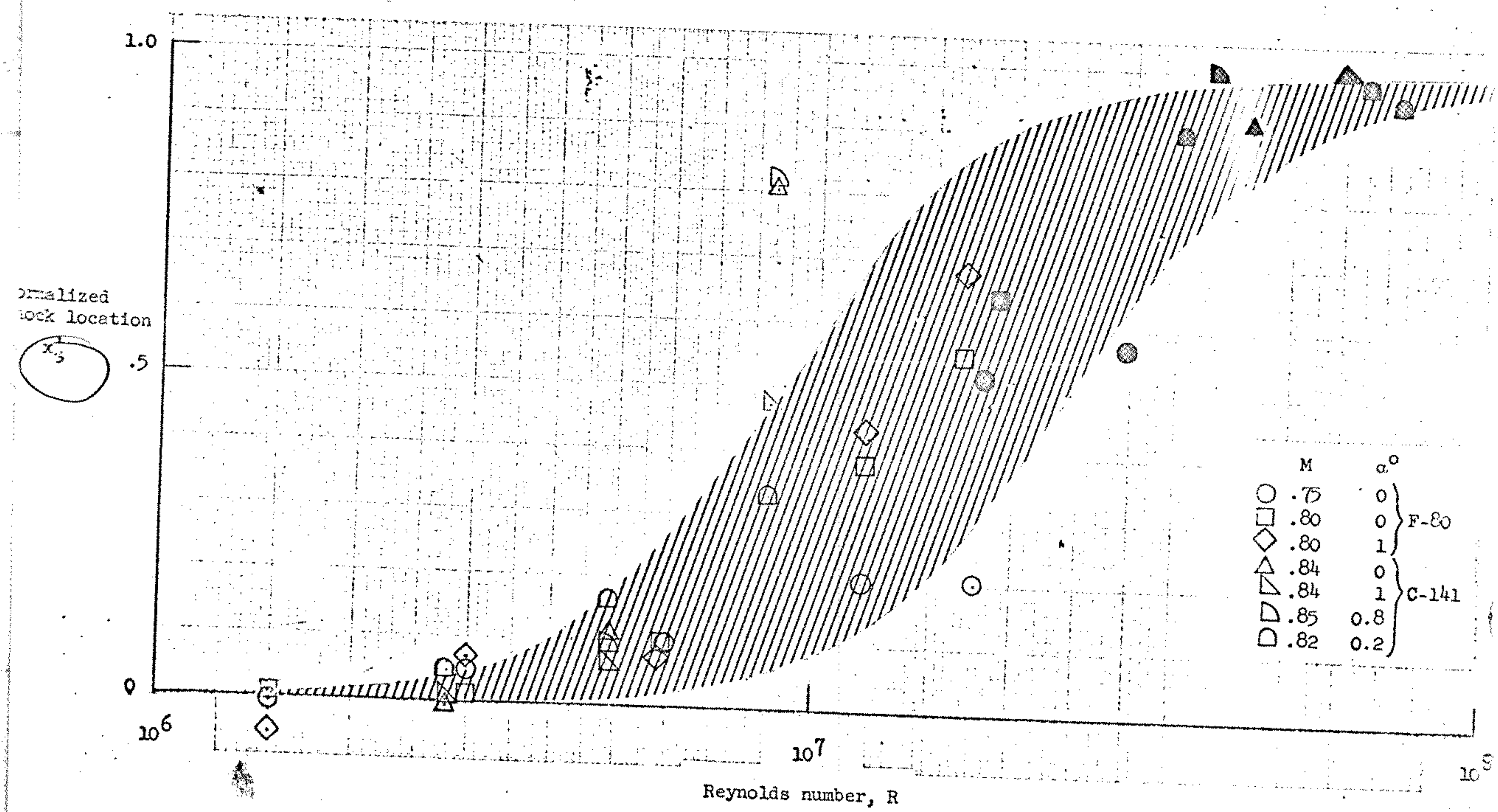


Figure 5.- Variation of normalized shock location with Reynolds number (solid symbols indicate flight data).